C3D – an imaging radiation damage experiment on UKUBE-1

C3D – AN IMAGING RADIATION DAMAGE EXPERIMENT ON UKUBE-1

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ABSTRACT

The Open University, in collaboration with e2v technologies Ltd and XCAM Ltd, have been selected to fly an experimental payload on board the UK Space Agency's UKube-1 pilot Cubesat programme. Cubesat payloads offer a unique opportunity to rapidly build and fly space hardware for minimal cost, providing easy access to the space environment. The proposed payload incorporated new imaging technology using a CMOS image sensor into a combined Earth Observation (EO) technology demonstrator and in-orbit radiation damage characterisation instrument, to help raise the TRL of the sensor technology. Based around the e2v 1.3 MPixel 0.18 micron process “eye-on-Si” CMOS devices, the instrument consists of 3 distinct image sensors; one devoted to radiation damage monitoring (RDM), as well as a narrow field imager (NFI) and a wide field imager (WFI). The narrow and wide field imagers are expected to achieve resolutions of 25 m and 350 m respectively from a 650 km orbit, providing sufficient swathe widths of 30 and 450 km respectively. The radiation damage experiment has been designed to verify and reinforce ground based testing that has been conducted on the e2v eye-on-Si family of devices and includes a TEC for temperature control as well as RADFETs for in-orbit dosimetry. Of particular interest are Single Event Effects (SEEs); Single Event Upset (SEU) and Single Event Latchup (SEL) effects etc. and the experiment contains operating modes to evaluate these during SAA passage. The novel instrument design allows for a wide range of capabilities the within highly constrained mass (170g), power (1W) and space budgets providing a model for future use on similarly constrained missions, such as planetary rovers. Scheduled for launch in June 2014, this project should not only provide valuable data helping to raise the TRL of the technology to prove flight heritage for future missions, but also provide outreach opportunities demonstrating the capabilities of such payloads.

1 INTRODUCTION AND OBJECTIVES

e2v are established suppliers of CCD technology into spacecraft (such as XMM, Gaia, Hubble, MarsExpress etc.) however their newer CMOS image sensors have not to-date been flown. In response to a call from the UKSA for payload concepts for their first national CubeSat programme called UKube-1 [1], we proposed an imaging payload which would demonstrate e2v’s new CMOS integrated imaging sensors which will:

- be the first demonstration of a European CMOS imager using 0.18 μm-technology in space
- be the first demonstration of an e2v CMOS imager in space

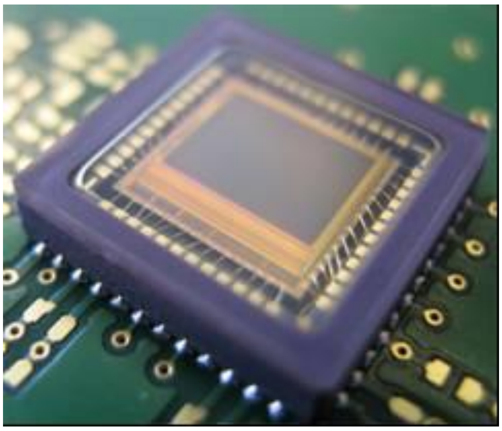
- be a technology demonstrator for use of the new range of CMOS imaging technology being developed by e2v for other mission opportunities, such as the JANUS camera on JUICE, Solar-C, lunar and Martian rovers, and for Earth observation instruments etc.
- to increase the TRL of the technology
- to demonstrate the technology for future missions and attract inward investment from other programmes to develop the technology further
- take high quality images of the Earth
- monitor the imager performance as the radiation damage increases (for both ionising dose and displacement damage) and to correlate this to laboratory measurements
- operate through SAA passages to evaluate single event effects

The project builds upon the existing knowledge exchange collaboration between e2v technologies and the Open University embodied in the Centre for Electronic Imaging, or CEI (www.open.ac.uk/cei) where one of the core research themes is the investigation of space radiation damage on image sensor technology. The imager is based upon e2v's new 1.3 Mpixel "camera on a chip" CMOS image sensor. Technology development and readiness is already advanced: radiation hardness and space applications have been investigated through an existing PhD studentship [2,3]. The device technology has been demonstrated to withstand 10^{10} protons.cm^{-2}, a TID of 200 krads and has been recently tested using heavy ions for SEU/SEL effects and has been demonstrated to work without any catastrophic burnout effects.

We called the payload C3D – a Compact CMOS Camera Demonstrator, which of-significance includes 3 CMOS image sensors, each performing different functions; wide and narrow field imaging, plus radiation damage assessment. Being a radiation damage experiment, which also takes images, the instrument also contains thermometry and dosimetry monitoring functions to enable the results obtained to be correlated with those obtained in laboratory testing on the ground. The following sections describe the rationale behind the instrument, the CMOS image sensors, and the C3D payload in greater detail.

2 e2v CMOS IMAGE SENSOR

2.1 Sensor Description



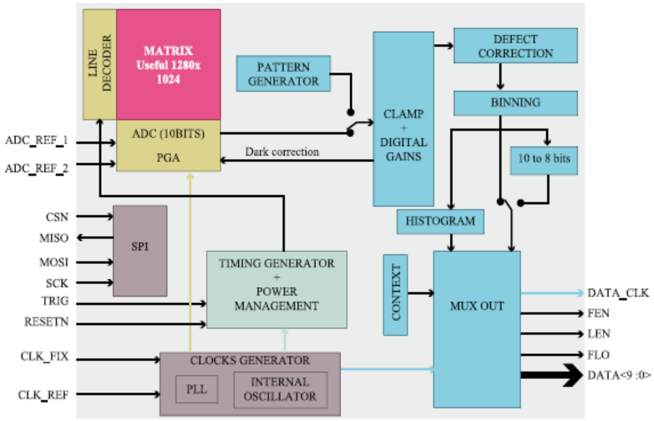


Figure 1 : The 0.5 Mpix CMOS imager used in the previous radiation damage test campaigns (left) and the system architecture of the larger 1.3Mpix variant (right). This highly integrated sub-system makes the creation of a space imager with much lower mass/power/volume requirements than using more conventional CCDs

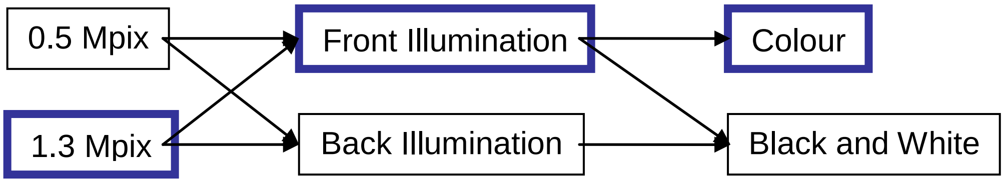


Figure 2 : Imager options available for the instrument at the time of the proposal, and in blue the baseline option chosen (note that a 2 Mpix version of the sensor now exists also)

e2v is a world-leading provider of CCD-based scientific imaging sensors into the global space community, specialising in high performance sensors, customised to the application needs, often with back-illumination to improve sensitivity. In the future, this tailored service to provide customised sensors will extend to the provision of application-specific CMOS imaging technology; indeed some missions are already exploring custom designs (e.g. Earthcare). In this instrument we proposed taking one of the new “Eye-on-Si” CMOS imagers [4] and providing the first flight demonstration of the technology in space. This instrument is undoubtedly *not* be the first flight of a CMOS imager on a Cubesat, however the other imagers have been very simple COTS components. The **unique** aspect of this instrument is that we aim to fly one CMOS imager of a family whose design heritage will provide traceability from our existing space radiation damage qualification campaign, through to future space imagers coming from the e2v CMOS design stable.

Several variants exist of the proposed CMOS imager represented schematically in Figure 2 above. Whilst the technological demonstration could be performed on any variant, as this is a pre-cursor to flight of this technology on science missions, we might prefer to fly a back-illumination option for enhanced sensitivity. However, the tight timescales for original UKube-1 schedule precluded this. The key remaining variants to select between are therefore based on pixel number and colour vs. black and white. For the instrument design we chose the larger 1.3 Mpixel imager with its colour variant, which we believe will provide a greater return for this small technology demonstrator.

This concept builds upon the CMOS imager developments and space radiation damage effects which are two key research themes of the CEI. We have been involved with e2v’s CMOS imager developments since its humble beginnings in 2005 through UK CASE PhD studentships, and currently have 3 PhD students performing research into CMOS imagers exploring both their design and use, and radiation hardness for use in space.

2.2 Laboratory Radiation Damage Studies on CMOS Imagers

The CEI and e2v have been working for over 9 years on the development of new CMOS imager technology [2,3,5], and one student performed a detailed characterisation of the e2v ‘Jade’ 0.5 Mpix 5.7 μm pixel integrated CMOS sensor from the same family as that chosen for this work. This sensor has been characterized for proton damage effects up to $1x10^{10}$ cm^{-2} and using gammas to evaluate total ionizing dose effects with operational doses beyond 100 krads [2], and very recently has been evaluated for SEU and SEL effects using a heavy ion beam. The generation of dark current in these sensors with TID is given in Figure 3, showing a rapid rise in dark current generation with increasing dose. Figure 4 gives the increase in bright (and flickering) pixels with proton fluence. The conclusions from this work state that the radiation effects induced in the imager are broadly similar to that measured on existing CCDs (flatband voltage changes, dark current increase, the creation of bright and flickering pixels), which imply that for scientific use such imagers will still require cooling.

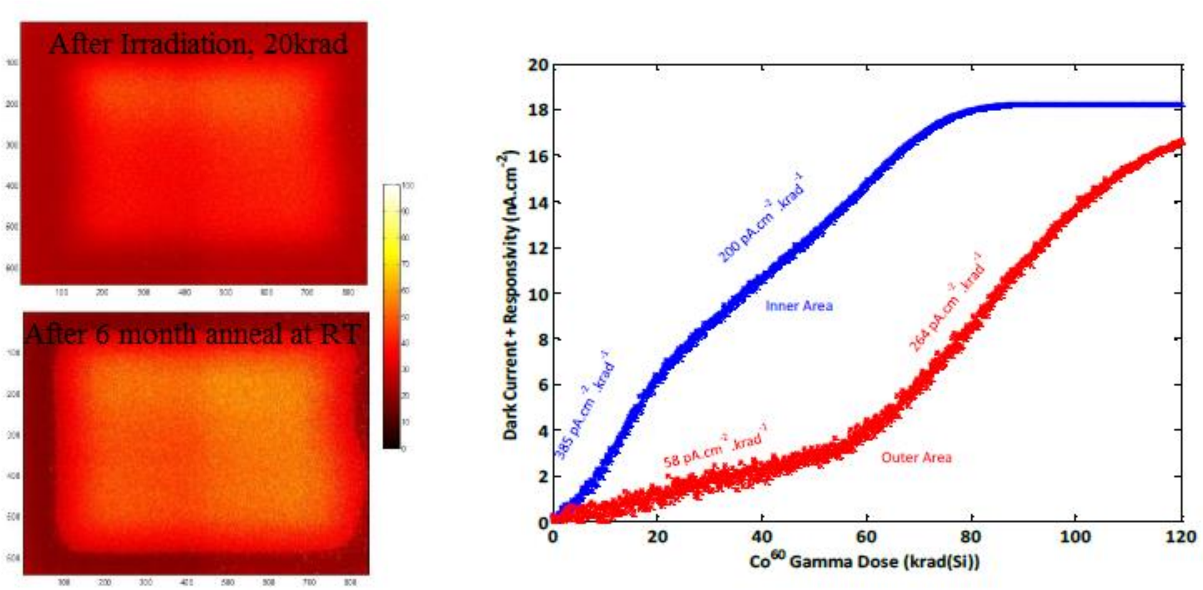
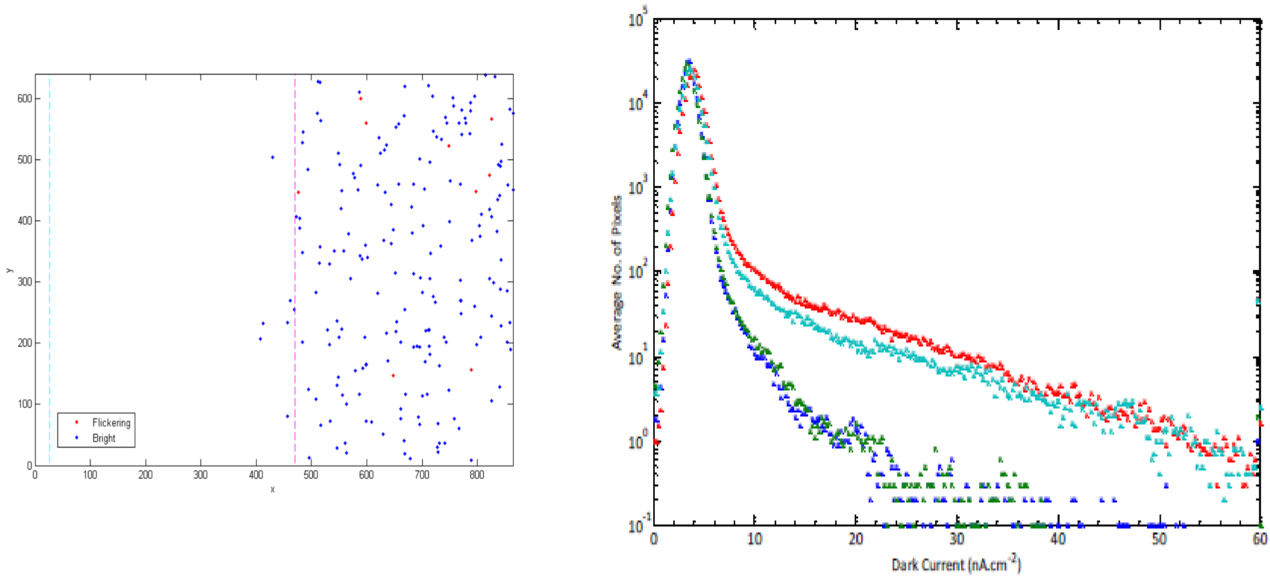


Figure 3 : Dark current generation in the e2v EV76C454 “Jade” with TID, including evidence of modest reverse annealing after storage.



a) b)

Figure 4 : a) Dark current spikes in the 76C454 “Jade” CIS irradiated over half its area and b) the dark current histograms both before (blue+green) and after 1x10^{10} cm^{-2}, 10 MeV protons (red+turquoise) from [3]

The CMOS sensors clearly do not suffer the charge transfer losses experienced by CCDs, however they are susceptible to SEU and latchup effects which are not a concern for CCDs. Figure 5 shows the results of latchup testing conducted at HIF in Louvain as part of this instrument development on the EV76C560 “Sapphire” sensors. The sensor was operated for blocks of time and periodically re-set by the system. A latchup event produces an increase in current consumption, and the device becomes un-responsive, and holds this state until the next reset. Non-catastrophic latchup events were monitored which caused the device to stop functioning temporarily, however normal operation resumed following the next power-cycle of the sensor. Figure 5a shows several such latchup events particularly to the circuitry on the 1.8V bias line, and Figure 5b shows that the onset of significant device latchup occurs at LET values typically 16 MeV.cm^2/mg and higher. Significantly, of all of the devices tested (both Jade and Sapphires types), no catastrophic burnout occurred.

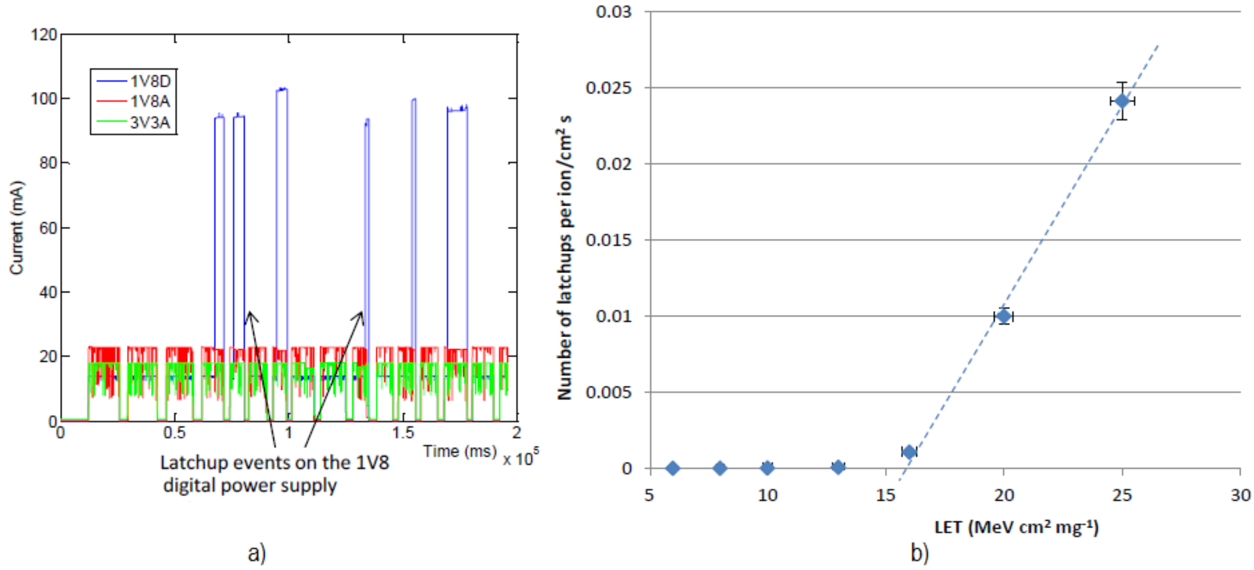


Figure 5 : Latchup in the Sapphires sensor with a) events monitored at 15.9 MeV.cm^2/mg in the 1.8V bias line to the sensor and b) latchup frequency vs. LET

One of the CubeSat experiment goals was therefore to explore these various radiation-related effects on-orbit and to enable a correlation of the observed effects with those measured on the ground.

3 C3D PAYLOAD CONCEPT

The instrument, designated the Compact CMOS Camera Demonstrator (C3D) was therefore conceived as a technology demonstrator, both demonstrating the imaging technology, and by designing a complex payload, to demonstrate some of the capabilities of the CubeSat platform.

3.1 Instrument Technical and Scientific Objectives

The instrument objectives are listed below:

- be a technology demonstrator for use of the technology on other mission opportunities, such as the JUICE imager, Solar-C, lunar and Martian rovers, and for EOS etc.
- monitor the imager performance as the radiation damage increases (for both ionising dose and displacement damage) and to correlate this to laboratory measurements
- operate through some SAA passages to evaluate single event latchup effects
- to monitor the instrument temperature profile over complete orbits using a low-power mode
- to provide accurate TID dosimetry at the instrument (and hence spacecraft) using RADFETs
- take high quality images of the Earth which can be used for outreach, schools, and showcase the CubeSat as an imaging platform
- Contribute to the training and research of at least 3 PhD studentships and PDRAs

3.2 In-Orbit Radiation Damage Assessment

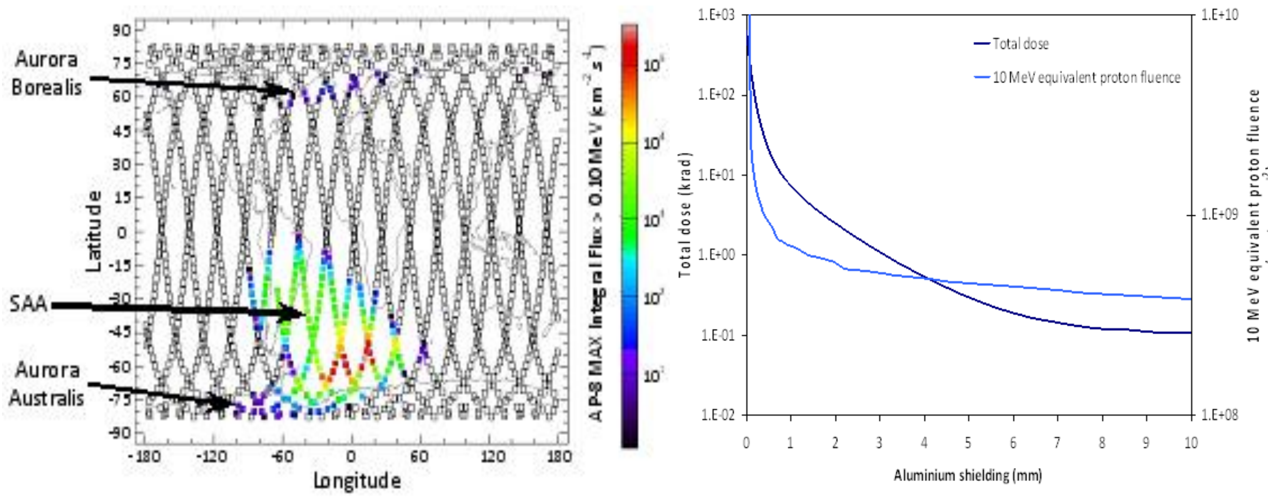


Figure 6 : Spenvis analysis of the baseline UKube orbit (650km, near polar) showing the SAA and horns, together with a plot showing total dose and proton fluence vs. shielding thickness from the radiation environment study

All silicon imaging sensors suffer increases in dark current during operation in space, and in CCD and CMOS imagers this can particularly manifest itself as an increase in bright pixels, some fraction of which, up to 50%, also flicker. One of the core competencies of the CEI is in space radiation damage effects, and one of the key aims of our instrument will be to enable tracking of the accumulation of radiation damage effects, particularly the growth in bright and flickering pixels, plus performing a study of single event effects and latchup during passage through the SAA.

Figure 6 shows the results of the orbit analysis for the baseline orbit at 650km altitude, with proton rate activity showing passage through the SAA and north/south horns. A simple spacecraft shielding model has been generated giving proton flux and TID vs. aluminium shielding thickness. This study enables us to estimate for the notional shielding provided by both instrument and the rest of the spacecraft that the imager might experience an annual ionising dose of ~1 krad/yr, and 10 MeV-equivalent proton fluence of ~3.5×10^{8} $cm^{-2}yr^{-1}$. The combination of device latchup testing on the ground plus the Spenvis modelling of the orbit leads to the estimate that the CMOS imager might incur a latchup event once every ~10 passages through the SAA. This data helps guide our estimates for the latchup experiment of the payload.

3.3 Earth Imaging

One of the goals of the payload was to take images of the earth using the CubeSat platform. We decided to provide two imaging systems on the payload, one using a standard camera lens with ~40^{o} FOV, designated the Wide Field Imager (WFI), and the second trying to push the limits of resolution, using a Cassegrain Telescope configuration with focal length ~150mm, designated the Narrow Field Imager (NFI). Figure 7 shows the main options for the imager (0.5 Mpix and 1.3 Mpix options) assuming a field of view (FOV~30^{o}). In red and pink the FOVs of the baseline 1.3Mpix images is given with a 10 s slip in image exposure which may arise due to the uncertainty in absolute exposure commanding for the spacecraft. In practice, it will be possible to collect a series of overlapping images and store in the instrument memory, e.g. with an ~25s interval to ensure capture of a specific/extended region. This mode would, however, place subsequent requirements on the very restricted telemetry budget for UKube-1.

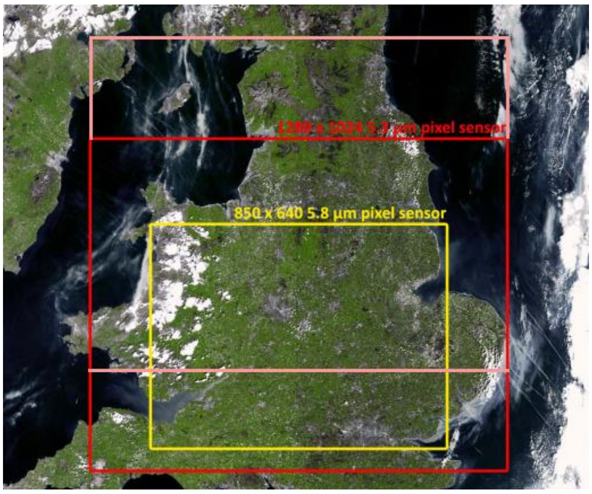


Figure7: Example views from the two sensor options for the WFI with simple lens at 650 km orbit. An example of the motion over the ground in a 10s interval is demonstrated for the larger sensor.

Using a standard lens from Sunex with 40^{o} FOV, the spatial resolution is ~350m from 650 km altitude. Using the on-chip global shutter option, the image exposure may be programmable from a few µs to several seconds with a typical exposure of the sunlit Earth being ~10 ms; equivalent to 70m motion over the earth. This will freeze any image blur due to the motion of the spacecraft when using such an optical arrangement. The typical exposure time will be set to ensure that bright scenes (e.g. cloud top) are sampled within the 10-bit dynamic range of the sensor. The longer exposure times are intended for use to evaluate the growth in dark defects in the imager as the mission progresses as indicated in Figure 4a.

Figure 8 : Example FOV with the NFI Cassegrain telescope with a pixel of 25m for a focal length of 15cm from 650 km orbit, together with an indication of such resolution over London. Taking such narrow field images will require good knowledge of the spacecraft pointing.

Besides using a standard lens for the WFI, we also wanted to attempt to perform medium-high resolution imaging using the CubeSat platform. To achieve this we conceived of using a Cassegrain telescope configuration, where a relatively long focal length could be achieved through the folded optic design. Due to the original constraints imposed by the spacecraft on our payload, we could only image along the short length of the 3U CubeSat, however we could achieve a focal length of

~15 cm, which would provide a pixel size on the ground of 25m. Examples of the typical FOV and resolution from 650 km orbit are given in Figure 8.

4 SYSTEM OVERVIEW

The C3D instrument uses the standard CubeSat PC104 experiment slot with a 9 x 9 x 2 cm^3 volume. The payload has several novel attributes. For the imaging, it possesses two CMOS image sensors. The first is behind a lens providing a wide field imager (WFI) of the Earth with an ~40^o FOV. The NFI has its own CMOS sensor and has an ~2^o FOV. The locations of the co-aligned NFI + WFI were constrained by the slots available in the standard Pumpkin 3U CubeSat support structure. The radiation damage in the sensor technology is monitored using a third sensor, creating the RDM. This third sensor is optically shielded from light so that the dark current can be measured. The RDM sensor is bonded to a thermo electric cooler (TEC) to enable the temperature to be stabilised for accurate repeatability of measurements. In addition, the RDM can be operated at elevated temperatures up to +80^{o}C to enhance the radiation-induced dark current artefacts. The experiment is complemented by inclusion of two RADFET dosimeters, each with their own PT100 temperature sensors. One RADFET is located on the experiment support system (ESS) PCB to provide an estimate of the TID received by the experiment. The other RADFET and PT100 is provided to the spacecraft to monitor other critical areas. In this instance the second monitoring position is by the spacecraft batteries. The two RADFETs will assist both instrument and mission teams to assess the accumulation of TID as the mission progresses.

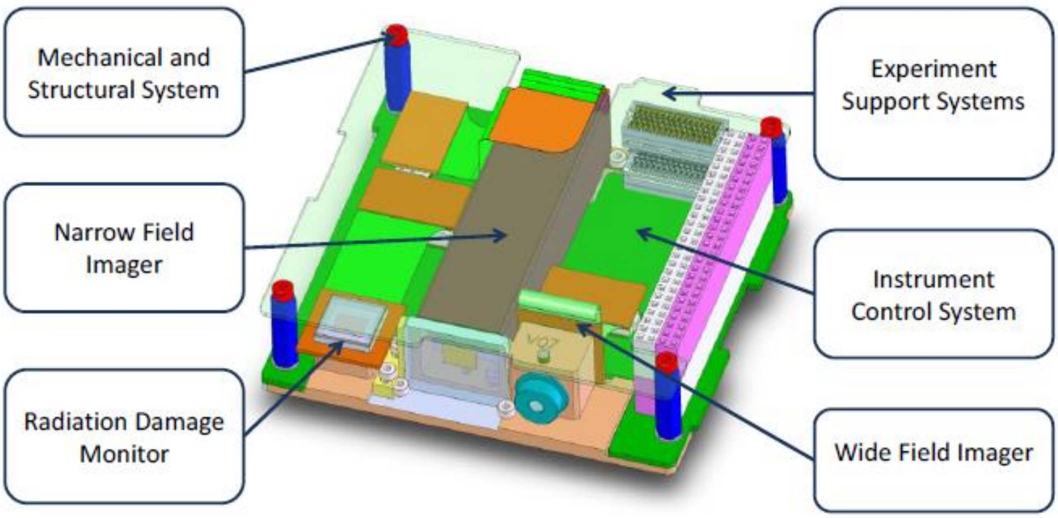


Figure 9 : Isometric CAD model of the C3D instrument showing the location of the main parts

Several models of the payload were assembled, and Figure 10 shows a photograph of one of the engineering models of the instrument with a ruler for scale. In this image the payload controller (top PCB) has an additional lower PCB which provides a simple USB interface to a laptop for ease of testing. The spacecraft manufacturers, ClydeSpace also provided a spacecraft emulator board which interfaced through the PC104 edge connector stack to enable full testing of the system prior to shipment.

Figure 11 gives an overview of the system functional blocks of the experiment and how it interfaces to the spacecraft. The two PCBs of the instrument can be seen; the Payload Control Electronics (PCE), and the Experiment Support System (ESS), in addition to the 3 CMOS image sensors which are controlled by the FPGA on the PCE.

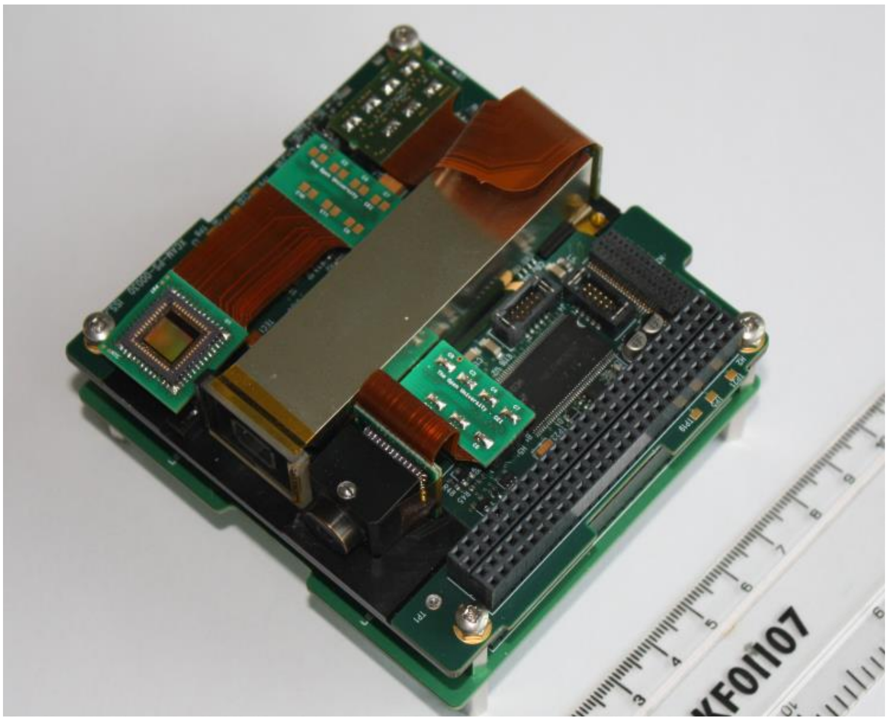


Figure 10 : Photograph of the assembled prototype C3D instrument

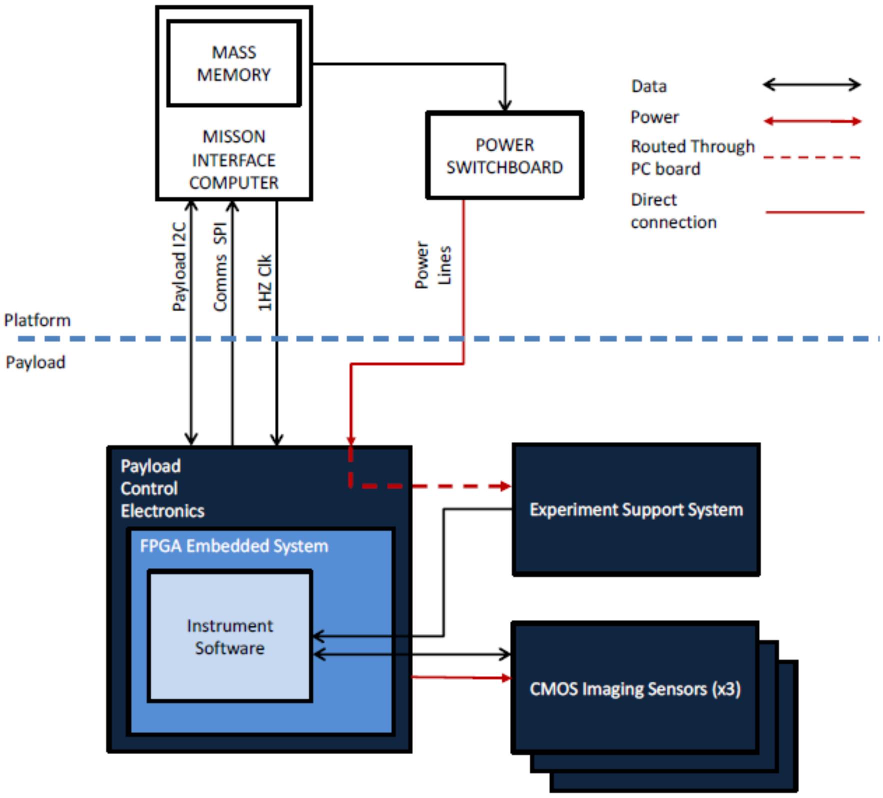


Figure 11 : Main components of the C3D instrument showing the payload control electronics (PCE) which drives the 3 CMOS sensors, and communicates with the experiment support system (ESS)

5 SUB-SYSTEM DETAILS

5.1 Payload Control Electronics - PCE

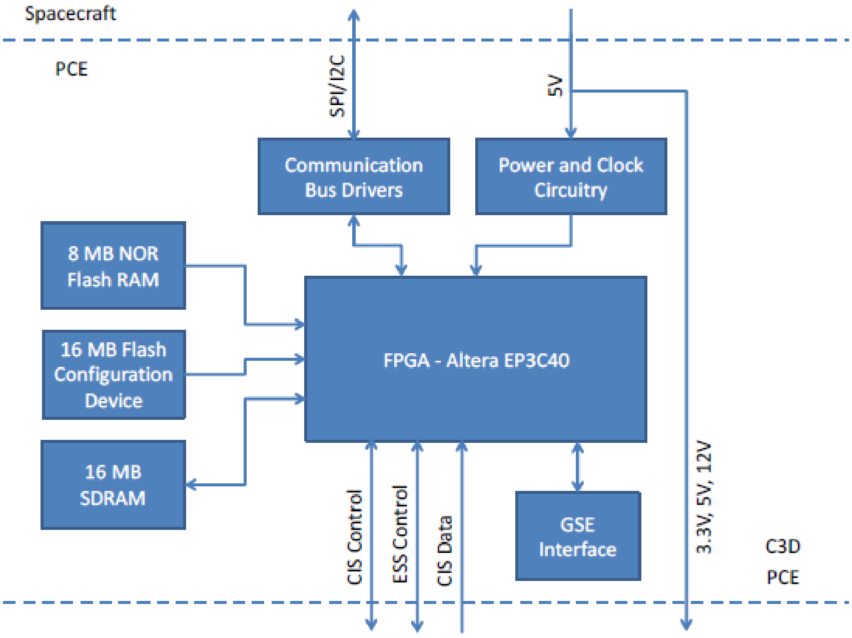


Figure 12 : Functional blocks within the PCE

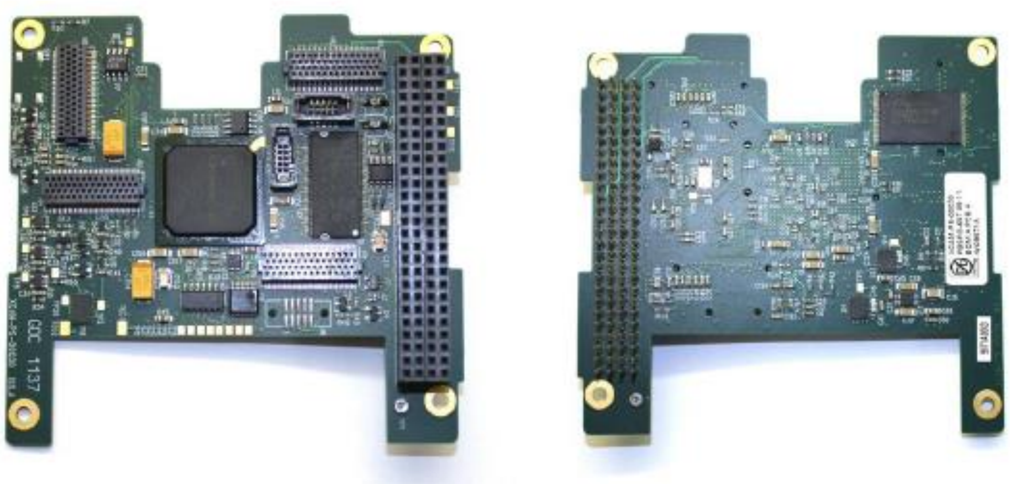


Figure 13 : Photographs of the payload control electronics which were developed by XCAM

The PCE was designed and manufactured by XCAM Ltd. Figure 12 gives a schematic of the main functional blocks within the PCE, whilst Figure 13 provides photographs of the top and bottom of the PCB. The PCE was constructed as a 10-layer PCB, and although the build standard could use COTS components, the assembly used Pb-solder to help suppress tin-whiskers which might develop over the 3-year lifetime on-orbit. The irregular shape to the PCB arose due to the constraints introduced by the mechanical structure of the instrument and the optical components. In particular, the experiment was built on an aluminium frame which was both used to support the optics and to provide a thermal heatsink for the power dissipated in the TEC of the RDM.

5.2 Experiment Support System - ESS

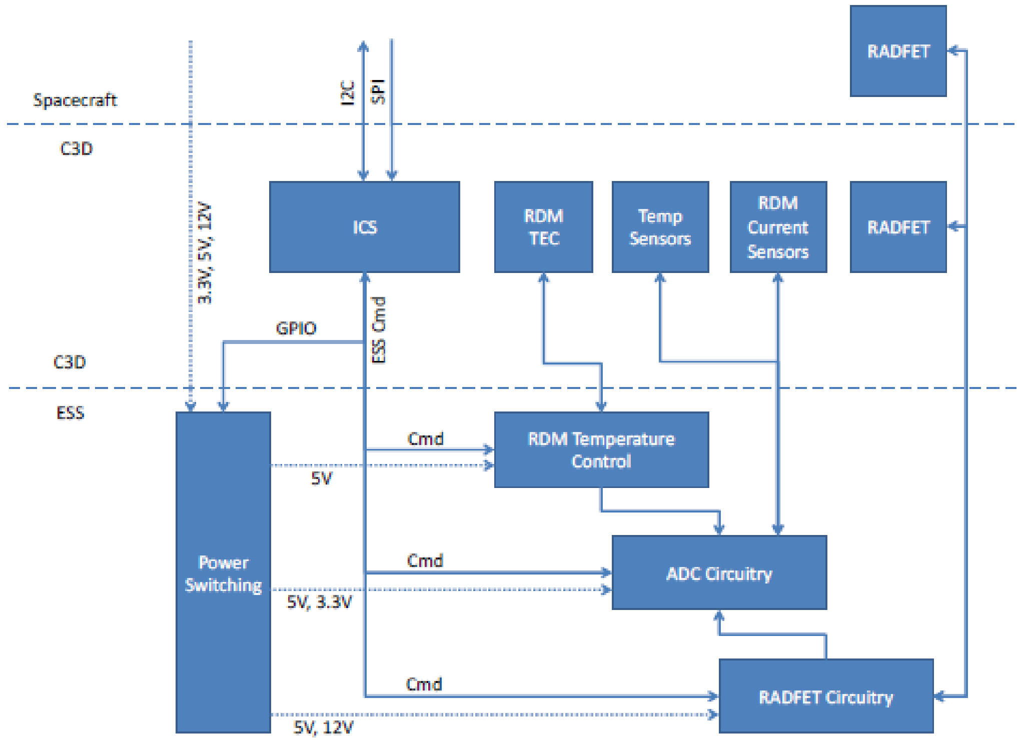


Figure 14 : Functional blocks within the Experiment Support System and their relationship to the RADFETs, temperature sensors and TEC

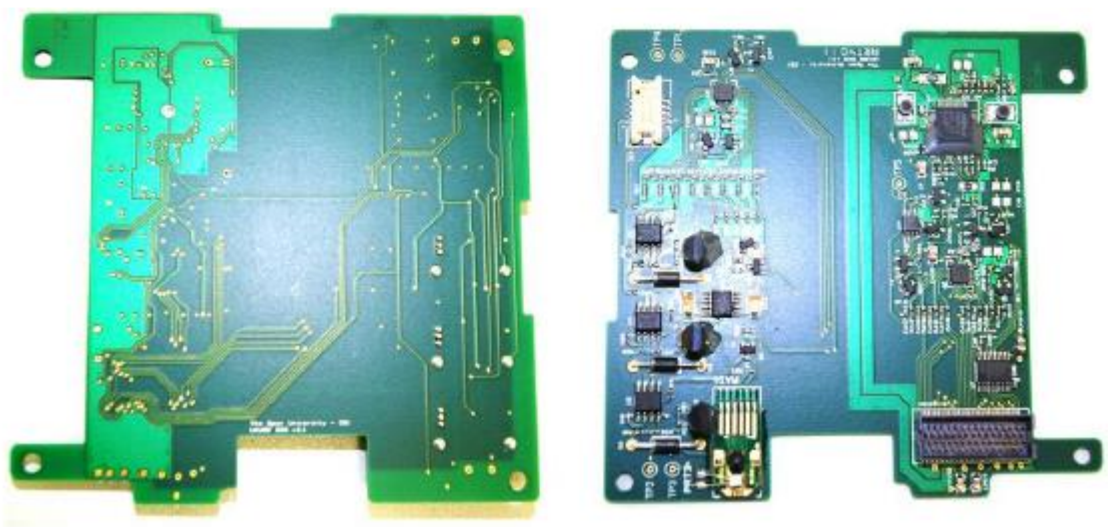


Figure 15 : Photographs of the Experiment Support System which was developed at the university

The ESS was designed and manufactured at the Open University. Similarly, Figure 14 provides a schematic of the main functional blocks of the ESS (lower half) and the connections to the RADFETs, PT100 temperature sensors, and the TEC. Figure 15 provides photographs of the top and bottom sides of the ESS PCB. The RADFET for monitoring the experiment TID can be seen in the lower left tab of the top side (right photograph). The relatively thick tracks are to handle the current for supply to the TEC. Again, the PCB cutouts were to accommodate the optics and the PC104 connector carrying the spacecraft bus.

5.3 Narrow Field Imager

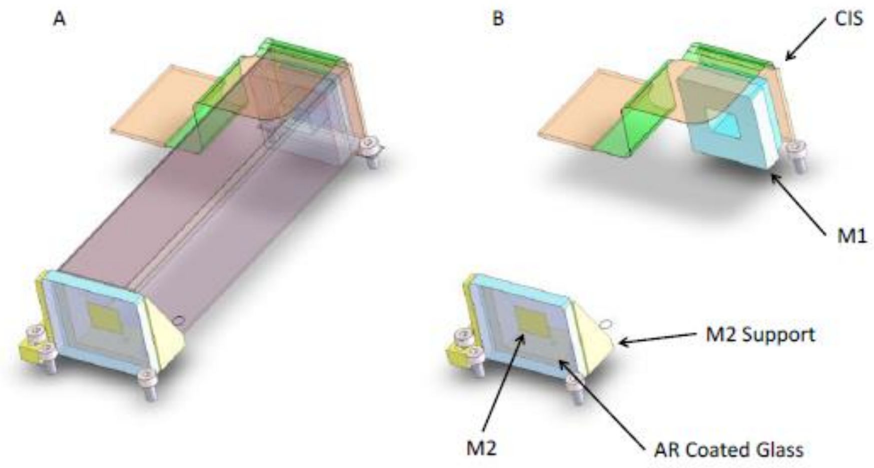


Figure 16 : Detail of the NFI telescope design showing a) the full construction including optical baffle and b) the mirror + sensor configuration

Figure 16 shows the design of the Cassegrain telescope forming the NFI. This is a 2-mirror configuration with M1 formed by precision diamond turning, whilst M2 is formed as a reflecting zone on the main entrance window. The M1 and M2 are silver-coated for increased reflectivity and performance against tarnishing. The entrance window is a double-sides AR coated optical glass substrate. The assembly has a number of baffles to reduce the stray light. The overall configuration results in an aperture of ~1 cm^2 with a focal length of 145 mm. The CMOS image sensor is bonded to the rear of the M1 behind the central hole in the mirror.

5.4 Mass, Power and Data Budgets

The instrument mass, power and data volumes are given in Tables 1-3 below. The electronics has standby mode, imaging and housekeeping (when temperature and RADFETS are read), and each of these modes has slightly different power consumptions. When operating the RDM with thermal control using the TEC, the power increases. It should be noted however that in normal use C3D will only be operational for 1-3 minutes to take the images and transfer the data to the main on-board memory. In addition, due to the telemetry budget, the transfer of a full resolution image, with compression, may take between 3-4 days. Because of this limitation, the PCE also creates thumbnail images of reduced size. These thumbnails can be downloaded in a couple of hours to enable a decision to be made on which images to download and which, if any, to discard. The very low duty cycle imposed on the instrument due to the telemetry bandwidth implies that the orbit-averaged power will therefore be very low, in the few mW regime.

Table 1 : Power Budget

Power (mW)			
	Mean	Peak	Maturity
Standby	880	950	Measured
Imaging	885	900	Measured
RDM+Temp Control	1500	1800	Measured

Table 2 : Data Volumes

Image Data Volumes	
1 Image (raw)	1.3 MBytes
Image with compression	100-650 kBytes
Thumbnail	12 kBytes
Housekeeping	250 bytes

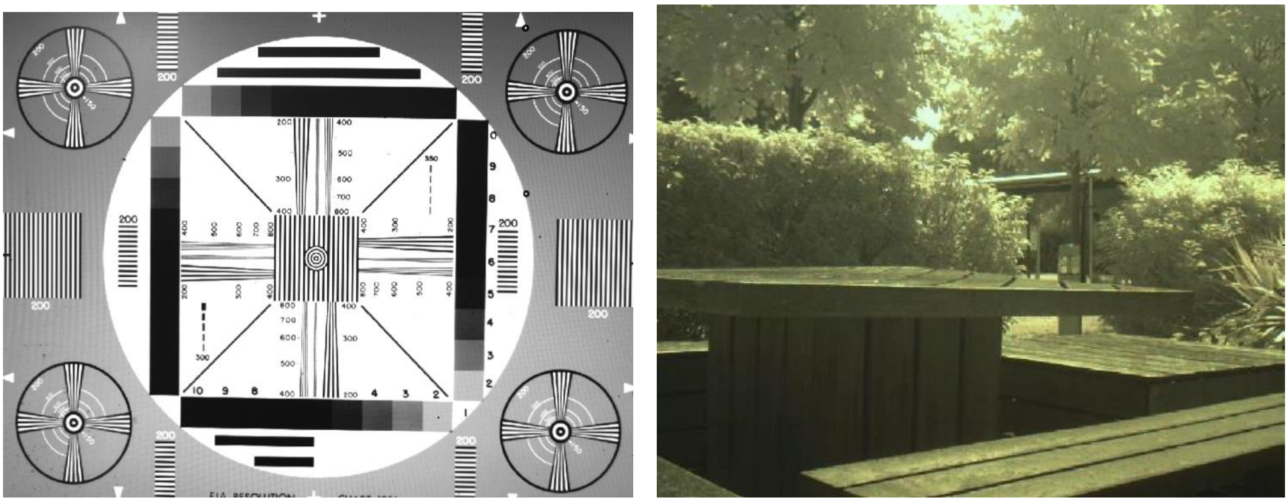
Table 3 gives the design mass breakdown of the instrument for the various component parts, together with mass margins. The main contributors to the mass of the instrument are the metal framework and heatsink for the TEC and the various connectors themselves. The measured mass of the final flight unit was 175g including all fixings, adhesives and mounting framework.

Table 3 : Mass Breakdown

Item	Mass (g)	Margin
Metalwork	62	20%
Cold Finger	1.5	20%
Connectors	40	10%
Fixings	5	20%
PCBs	21	20%
CMOS Imagers	2	5%
Lens+Mirrors	15	10%
TEC	2	10%
Total	**148 g**	**(180 g inc.)**

6 TESTING

6.1 Imaging



a) b)

Figure 17: Images taken using the WFI of scenes; a) of a standard test chart and b) from a scene outside the laboratory on a bright sunny day, using the automatic exposure routine with an illumination of 700 Wm^{-2}

Exposure control can take two forms in the payload; fixed and automatic. When using fixed exposure the exposure time can be specified as part of the grab command. Alternatively, the payload has an automatic exposure mode, where a succession of images are taken with longer and shorter exposures until an image is acquired where <5% of the pixels are saturated. The final exposure uses this setting. The exposure times can range from μs-s however the WFI and NFI will use the ms-ms range, mainly in automatic exposure mode, and the RDM will use the 100ms-s range using a fixed exposure time to enhance the radiation-induced dark current.

Figure 17 shows a scene taken outdoors with one of the engineering units with the automatic exposure which was reported as being 190 μs. The irradiance at the time was measured to be 700 Wm^{-2} which is within a factor ~2x of that expected in-orbit. Similar scenes have been taken of cloudscapes and on darker days. We are therefore confident that imaging parts of the payload will be able to handle the range of illumination conditions which will be encountered on-orbit.

6.2 Environmental Testing

The flight model was subjected to a light set of environmental tests, including when integrated on the spacecraft after delivery to ClydeSpace. The qualification model underwent a more rigorous testing including the following:

- Full vibrational testing
- Thermal cycling over the range -35^{o}C to $+75^{o}$C, 6 cycles
- Thermal soak test
- Thermal vacuum test
- Gamma TID test up to 10 krads.

In addition, the CMOS image sensor type was subjected to heavy ion testing reported above.

7 INSTRUMENT SUMMARY

Table 3 gives the key parameter summary for the C3D instrument.

Table 3 : Key Parameter summary for the C3D Instrument

Criteria	Performance	Notes
	Optical	
Optical system	NFI: Two mirror reflecting Casegrain telescope with AR coating on entrance window WFI: Doublet lens with attached UV and IR filter	Broabban AR coating with peak at 550 nm 450 nm and 700 nm UV and IR cut off on WFI
SNR	1000:1 (at full well)	Data reduced to 8 bits in flight configuration
Contrast	MTF of 50% at Nyquist for 600 nm (dominated by sensor MTF given that the optical system is not Rayleigh limited)	
Frame rate	Effective frame rate 2 frames/minute	Limited by data bus. C3D can capture 5 FPS and locally store up to 6 images ready for later download
Compression	Lossless JPEG-LS	50% compression ratio
Exposure control	Manual or auto	
Thumbnailing	100 x 120 pixel thumbnails produced	
Windowing	Image windows can be produced on board instrument	
	Electrical	
Bus Interfaces	SPI, I2C, UART	Utilises the UKube-1 communication protocol
Required supplies	3.3V, 5V, 12V	
Nominal power consumption	900 mW	
Application memory	16 MBytes SDRAM	
	Mechanical	
Size	Approx.: 94 mm x 92 mm x 26.5 mm	
Mass	175 g	
	Additional Features	
Dosimetry	2 RADFET dosimeters allowing measurement of up to 32 krads (Si) with a resolution of approx. 10 rads	One RADFET located on solar panels of spacecraft to provide reference data
Temperature control	Control of RDM to within 1 °C between -20 and +70 deg C	

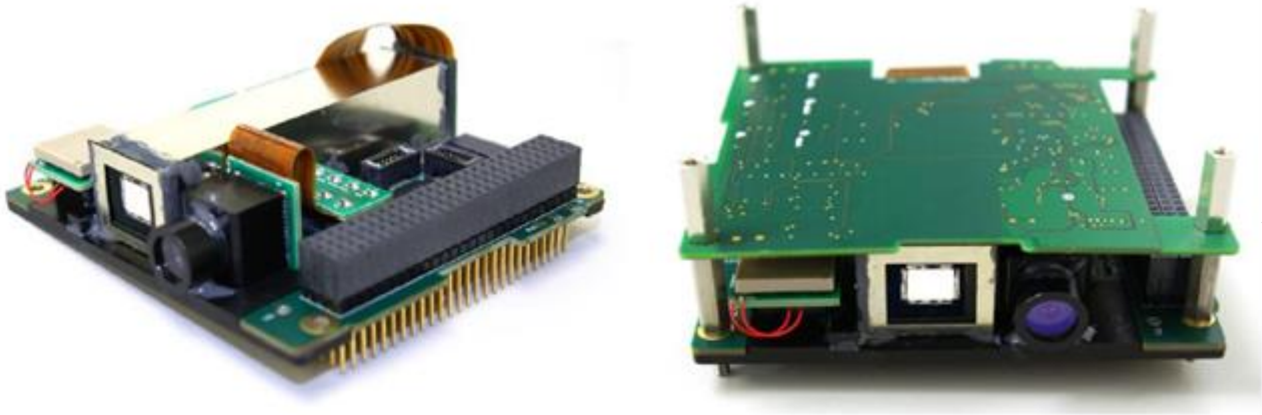


Figure 18 : Images of the C3D flight model prior to shipment, with the optics, sensors and control system on the left, and fully assembled with the ESS on the right. The completed size is 90x90x20 mm^3 with a mass of 175g.

8 CONCLUSIONS

We have described the rationale behind and design of a compact CMOS camera demonstrator for a CubeSat payload, the flight version of which is depicted in Figure 18 above. The instrument is comprised of an experiment controller with three separate CMOS image sensors which perform different functions; a radiation damage monitor, a wide field imager and a narrow field imager. In addition, the payload forms the basis of a radiation damage experiment and using PRTs and RADFETs can measure temperatures and radiation doses at specific locations, and can test for single event effects in the CMOS sensors themselves. On-board UKube-1, with a current launch date of 19th June, and a possible operational lifetime of 3 years, we look forward to a wealth of technical data from the payload, allowing in-orbit demonstration of the new imaging technology and confirming radiation testing on the ground. We also envisage many images from the combined wide and narrow field optics, to further demonstrate the capabilities of the CubeSat platform.

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